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APOLLO EXPERIENCE REPORT - ENVIRONMENTAL ACCEPTANCE TESTING

Charles H. M. Laubach

Lyndon B. Johnson Space Center
Houston, Texas 77058

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16. Abstract

Environmental acceptance testing was used extensively in the Apollo Program to screen selected spacecraft hardware for workmanship defects and manufacturing flaws. The minimum acceptance levels and durations and methods for their establishment are described in this report. Component selection and test monitoring, as well as test implementation requirements, are included. The Apollo spacecraft environmental acceptance test results are summarized, and recommendations for future programs are presented.

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By Charles H. M. Laubach
Lyndon B. Johnson Space Center

SUMMARY

The Apollo environmental acceptance test program is described in terms of the test background at the outset of the Apollo Program, the experience gained from vibration acceptance testing, the introduction of thermal/thermal-vacuum testing, the environmental acceptance test requirements, the implementation of environmental acceptance testing in the Apollo Program, and the results of this test program. Appendixes provide summaries of industrial surveys conducted on acceptance vibration testing and thermal/thermal-vacuum testing.

The environmental acceptance test program for the Apollo spacecraft resulted in the verification that the hardware, as manufactured, was adequate for flight before spacecraft installation. This test program proved to be an effective method for disclosing workmanship and manufacturing flaws. Regardless of how well the inspection procedures and functional tests were developed, environmental exposure of the hardware was found to be the best means of detecting many types of faults.

INTRODUCTION

The environmental acceptance test program consisted of three types of testing: vibration, thermal cycling in ambient conditions, and thermal cycling in a vacuum. The basic philosophy of the acceptance testing program was to provide the assurance that a given piece of hardware would perform reliably. A comprehensive test program includes qualification and acceptance tests. The qualification tests are designed to evaluate the hardware and to demonstrate that the hardware, as designed and manufactured, will perform as specified. The adequacy of the manufactured flight and test hardware can be verified through the acceptance test program. These tests ensure that the hardware is equal in quality to the qualification hardware.

Generally, qualification tests were conducted on one or two production articles, whereas environmental acceptance testing was conducted on all flight and ground test articles after the component types were selected for the environmental acceptance tests. The environmental acceptance tests provided verification that workmanship defects and manufacturing flaws, which could not be readily detected by normal inspection techniques, were not present in flight and test hardware. The environmental acceptance tests provided further verification that the quality of the hardware was acceptable for flight before installation in the spacecraft.

As an aid to the reader, where necessary the original units of measure have been converted to the equivalent value in the Système International d'Unités (SI). The SI units are written first, and the original units are written parenthetically thereafter.

ENVIRONMENTAL ACCEPTANCE TEST BACKGROUND

At the outset of the Apollo Program, a one-time qualification of a component or system design was performed. The qualification provided a reasonable margin of safety for the expected environments that the hardware would experience during storage, transportation, handling, and ground tests over two mission duty cycles.

At that time, it was proposed that a rigorous qualification program was not adequate in itself to provide flight quality hardware, and that each flight item should be subjected to some environmental testing as a part of acceptance. Although most functional components and systems underwent acceptance testing, the detailed test plans were left to the individual designers and systems engineers. Most testing was limited to functional bench tests at room temperature and pressure. A few components received a functional test after a brief exposure to vibration. This vibration was applied to the equipment in the most sensitive axis and at various vibration levels up to the expected flight-vibration environment. A few electronic component vendors, who were experienced in critical military programs and in other NASA programs, performed temperature limit tests at their own discretion during buildup or during final acceptance testing.

The first contractual attempt to impose specific environmental acceptance test requirements was in November 1965. These requirements were to have been implemented on the Block I command and service module (CSM) but were canceled in May 1966 because the Block I vehicles were in an advanced stage of assembly, and removal from the spacecraft of components requiring acceptance testing would have been necessary. The requirement was placed on the Block II spacecraft in February 1967.

The November 1965 acceptance test requirement was a random vibration excitation of 60 percent of the qualification power spectral density test level, but not less than 0.005 g^2/Hz for a minimum of 1 minute. The industry was surveyed regarding the philosophy and implementation of vibration requirements for acceptance testing so that inordinate requirements would not be imposed on the contractor. The results of the survey are discussed in the following paragraphs.

U.S. Air Force Programs

The U.S. Air Force required acceptance vibration testing on a majority of its hardware. Both random and sinusoidal vibrations were required at test levels representing the flight levels and from 3 to 6 decibels below the qualification level. In addition to other U.S. Air Force requirements, the first stage of the Titan III launch vehicle was static fired. This firing essentially subjected the hardware to a vibration test at the maximum environment.

NASA George C. Marshall Space Flight Center

The NASA George C. Marshall Space Flight Center had no formal requirement for acceptance vibration testing on Saturn launch vehicle hardware; however, some hardware did receive acceptance vibration testing. Each completed stage of the vehicle was static fired, which subjected the components to some vibration before flight.

Gemini Program

Gemini components as well as the complete spacecraft were subjected to acceptance vibration tests before flight. Components were tested throughout the program, whereas vehicle testing was discontinued after the third spacecraft. The vibration levels were 75 percent of the qualification level.

Industrial Practices

An industrial survey conducted by the Aerospace Industries Association of America (AIAA)[1] indicated that 80 percent of the companies surveyed used acceptance vibration tests. The average level used during testing was 60 percent of the qualification level. A total of 91 percent of the responding companies recommended acceptance vibration tests.

Whether uniform criteria had been applied to acceptance vibration testing of flight hardware by the contractors was not known. The extent of the nonuniformity of the CSM acceptance vibration testing was determined by evaluating acceptance test plans, procedures, and control drawings. Of the 415 hardware items, 303 did not receive an acceptance vibration test. The hardware items that were vibration sensitive and those that experienced failures during qualification vibration testing were delineated on a master list. This list contained many items that had not been subjected to vibration acceptance testing, further emphasizing the need for an adequate vibration acceptance test program.

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[1] Aerospace Industries Association of America: Industry Practices. Published in an AIAA letter signed by P. E. Everett, executive secretary, Nov. 10, 1966.

In early 1967, after the Apollo fire, spacecraft acceptance test practices were reviewed extensively. A questionnaire survey of Apollo subcontractor and vendor acceptance testing was conducted. The questionnaires included 79 questions concerning the subcontractor and vendor acceptance test plans and objectives. To secure a representative sampling of the varied technologies, 21 CSM and 12 lunar module (LM) components were selected for the survey. This survey revealed the inadequacy of environmental acceptance tests and, in many cases, their nonexistence. The vibration acceptance test levels were often based on the expected flight levels. Unfortunately, many of the expected vibration levels were so low that the early environmental acceptance tests did not reveal errors in workmanship and manufacturing processes. However, many of these faults were discovered later in the spacecraft checkout cycle; this situation delayed the program and resulted in the use of excessive manpower. Acceptance test environments must be severe enough to detect faults, yet not so severe as to weaken or fatigue the hardware to the point of reducing its useful life. In recognition of the generally too low or nonexistent spacecraft environmental acceptance test levels, an effort was undertaken to establish new levels and requirements for the Apollo Program.

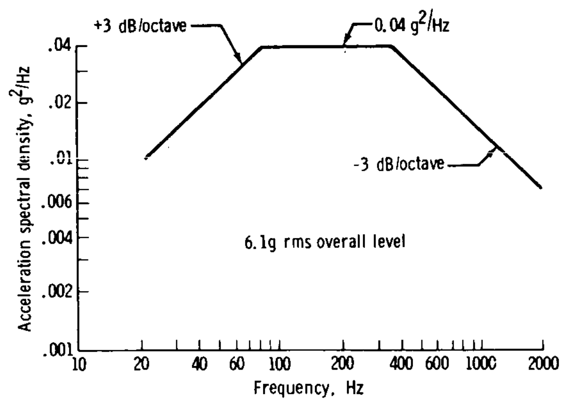
VIBRATION ACCEPTANCE TESTING

The study of early Apollo acceptance and qualification vibration failures revealed that workmanship and manufacturing faults not detected by the 3.5g to 4g root mean square (rms) levels during acceptance tests were later revealed by the 7.8g rms qualification levels. Early in the Gemini Program, acceptance levels slightly higher than 4g rms were imposed before the qualification testing of a component. This relatively low acceptance level (early Gemini acceptance program) permitted one of every two quality faults to enter the qualification program, whereas the levels used in the early Apollo Program permitted two of every three such faults to enter the qualification program. At the beginning of the Gemini flight program, the vibration acceptance level was raised to 6.2g rms, and 45 additional quality faults were screened from the previously acceptance-tested flight hardware; some of these could have resulted in critical failures during the mission. From the data, it was apparent that there was a threshold level below which many quality faults would not be detected. Also, the data indicated that the nominal threshold or minimum acceptance level should be established at approximately 6.0g rms.

Environmental exposure was used more extensively for acceptance testing in the successful unmanned spacecraft programs. Also, the levels used were much higher than those used in the Apollo Program. For instance, thermal vacuum and vibration were used for acceptance testing of the Mariner IV spacecraft. A 9g rms vibration level was used for acceptance testing, and a 16g rms level was used for qualification testing.

Based on the data obtained from the assessment of the Gemini experience and the other spacecraft programs, a more rigorous acceptance vibration test program was instituted on Apollo spacecraft components. A level of 6.1g rms and the spectrum shown in figure 1 were adopted as the Apollo spacecraft minimum acceptance vibration level. This shape spectrum was selected because the qualification tests for many CSM components were conducted to it and at 1.6 times this level, which was considered satisfactory.

The new vibration acceptance test requirements were contractually imposed on the CSM contractor in July 1967 and on the LM contractor in September 1967. There was concern about the wide variation in the acceptance vibration test requirements among the NASA centers and programs. The NASA Lyndon B. Johnson Space Center (JSC) (formerly the Manned Spacecraft Center (MSC)) conducted a survey to better understand the variations and to provide additional confidence in the new acceptance vibration requirements. A summary of this industrial survey is provided in appendix A.



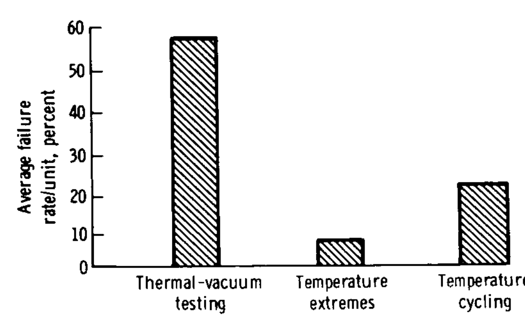
Note: The duration was a minimum of 30 sec/axis with an optimum of 1 min/axis. However, the duration was to be long enough to perform the required functional and continuity checks of all circuits during the test.

Figure 1. - Acceptance vibration test minimum level and duration.

The survey revealed that, in other programs, it was considered necessary to use environmental exposure as an acceptance criterion, regardless of how well the inspection and functional procedures were developed. Whereas the survey primarily gathered information on acceptance vibration testing, it also revealed that the most effective quality acceptance tool environments are vibration, thermal, and thermal vacuum. In many instances, the hardware was exposed to vibration and thermal or thermal vacuum; however, vibration alone was more often imposed.

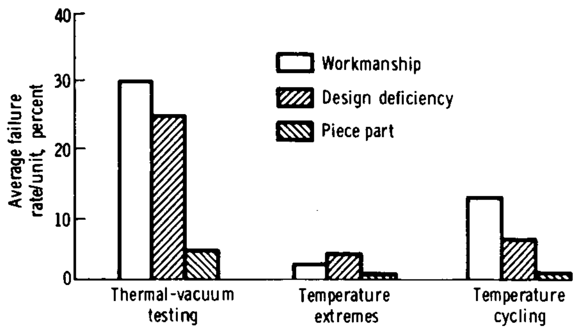
THERMAL/THERMAL-VACUUM ACCEPTANCE TEST

Environmental acceptance test data showed that, for many hardware types, vibration alone was insufficient for detecting some types of workmanship and manufacturing defects. Thermal and thermal-vacuum practices used on other programs, as well as early (pre-1968) LM environmental acceptance testing practices, were evaluated to establish uniform requirements to be imposed on the Apollo spacecraft hardware. The industrial survey conducted on thermal and thermal-vacuum acceptance testing is summarized in appendix B, and figures 2 to 4 contain data from pre-1968 LM environmental acceptance testing practices. The basic thermal/thermal-vacuum requirements adopted in May 1968 for the Apollo spacecraft hardware are shown in figure 5.



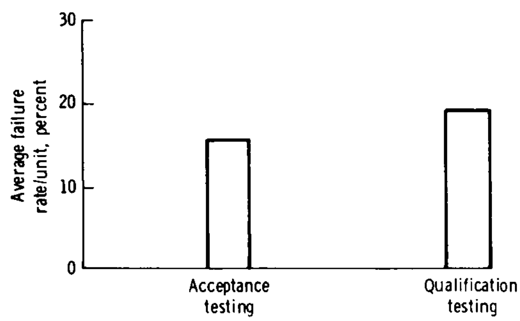
(a) Occurrence for each thermal test type.

Figure 2. - Acceptance test failures during thermal testing of LM hardware (pre-1968).



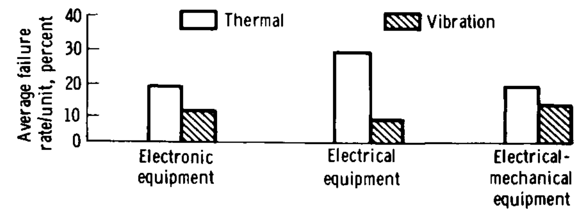
(b) Failure causes.

Figure 2. - Concluded.



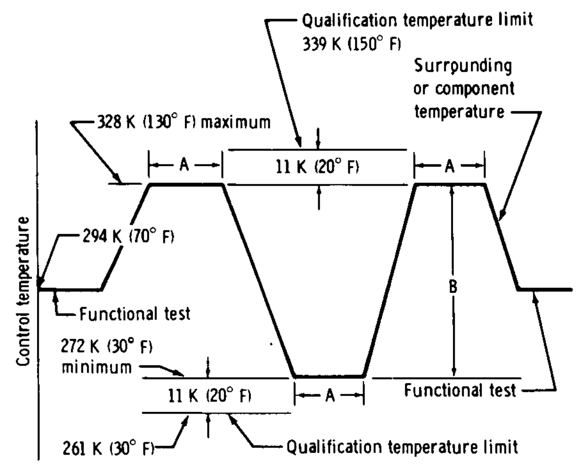
Note: The percentage reflected for qualification testing includes only that equipment tested in the thermal or thermal-vacuum environment.

Figure 3. - Qualification and acceptance test failures during thermal and thermal-vacuum testing of LM hardware (pre-1968).



Note: Thermal testing includes thermal vacuum, temperature, and temperature cycling. Vibration testing includes random and sine.

Figure 4. - Comparison of thermal and vibration failures during environmental acceptance testing of LM hardware (pre-1968).



A - Time to stabilize equipment temperature plus 1 hour minimum

B - The acceptance test control temperature range between the maximum and minimum test conditions should be a minimum of 56 K (100° F).

Note: Equipment was operated and continuity was monitored continuously with functional tests performed as shown at temperature extremes.

Figure 5. - Minimum requirements for component thermal cycle acceptance test.

ENVIRONMENTAL ACCEPTANCE TEST REQUIREMENTS

Acceptance testing included exposure to one or more environments, as required to detect possible faults. The following faults were expected to be exposed by acceptance vibration testing.

1. Loose electrical connections, nuts, bolts, etc.

2. Relay contact chatter

3. Physical contaminants
4. Cold solder joints and solder voids
5. Incomplete weld joints
6. Close tolerance mechanisms
7. Incomplete crimp connections
8. Wiring defects (i.e., strands cut away with insulation removal)
9. Shrinking of potting resulting in loose assembly within housing
10. Too soft potting permitting excessive movement of components and wiring

Faults expected to be exposed by acceptance thermal/thermal-vacuum testing are listed in table I. The number, duration, and severity of tests were not to cause overstressing or degradation of the capability of the hardware to perform its intended function. Where possible, all normal, alternate, redundant, and emergency operational modes were tested.

The acceptance tests were to be performed with strict adherence to the environments and test procedures. The hardware was calibrated and alined before acceptance tests were conducted. Adjustment or tuning of the hardware was not permitted during testing unless the adjustment was normal to the inservice operation.

For environmental acceptance testing, a failure was defined as the incapability of the component to perform its required function under the conditions and duration specified in the acceptance test specifications. After any repairs, modifications, or replacements during or after completion of acceptance tests, retesting was required to ensure the acceptability of the hardware. Retest requirements were to be proposed and submitted to NASA for approval.

A retest time limit was established for each type of component. A total acceptance test time, including the anticipated retest time, was established for each component and included in the qualification test requirements.

Hardware Assembly Level

A hardware assembly level was selected such that the dynamic transfer function of the structure caused a minimum magnification or damping of the input to the internal parts. Additional considerations were the assembly level of replaceable spares (black box level) and the capability of the assembly to be operated and monitored during testing.

TABLE I. - FAULTS EXPECTED TO BE EXPOSED BY ACCEPTANCE

THERMAL/THERMAL-VACUUM TESTING

Characteristic	Environment[a]				
	Thermal	Thermal cycling	Vacuum	Thermal vacuum	Vacuum cycling
Potting voids	X	--	X	(X)	--
Short run wires	X	(X)	--	--	--
Welded and soldered connections	X	(X)	--	--	--
Corona leakage	--	--	--	--	(X)
Outgassing contaminants	--	--	X	(X)	--
Bimetallic effects of leaf spring	(X)	X	--	--	--
Solder splash on printed circuits	--	--	--	(X)	--
Insulation penetration	--	(X)	--	--	--
Thermal grease application	X	--	X	(X)	--
Close tolerance mechanisms	X	X	--	(X)	--
Hermetically sealed components, environmental seals	--	--	--	(X)	--
Thermal interface integrity	--	--	--	(X)	--
Thermal control paint	--	--	--	(X)	--

[a]The environment most likely to expose a type of fault is indicated by parentheses.

Hardware Selection

Each component or subsystem for which a certification test requirement existed was a candidate for environmental acceptance testing. The following criteria were used to select the particular items to be subjected to environmental acceptance testing.

1. Items that could not be effectively inspected during manufacture or items the assembly of which involved processes that made quality control difficult (all electrical/ electronic and electromechanical components)

2. Items that had delicate mechanisms requiring precise adjustments

3. Items that had marginal environmental sensitivity

4. Items that were known to have high failure rates early in life

After a component type was selected for environmental acceptance testing, 100 percent of those flight and ground test items were tested.

Acceptance Vibration Test Levels and Durations

The vibration test levels and spectra were to the expected mission level or the acceptance vibration test minimum (fig. 1), whichever was greater. The test duration was a minimum of 30 sec/axis; 1 min/axis was considered to be the optimum duration. However, a functional and/or continuity check on all circuits had to be performed during the test, but this requirement seldom resulted in a test time of more than 1 min/axis.

Acceptance Thermal/Thermal-Vacuum Test Levels and Durations

The temperatures used for the dynamic thermal/thermal-vacuum tests were the expected mission level change from minimum to maximum or a minimum temperature sweep of 56 K (100° F) (fig. 5), whichever was greater. The vacuum level was 1.333 mN/m^2 (1×10^{-5} torr) or less. The test duration was a minimum of 1.5 temperature cycles with a functional or continuity check being performed on all circuits during the test.

Qualification Simulation

To ensure that the environmental acceptance testing had not degraded the hardware quality to the point of reducing its useful life, the qualification unit was subjected to testing with adequate margins to simulate the acceptance tests in addition to the normal qualification tests to cover the mission requirements. The qualification level to simulate acceptance vibration testing was defined as 1.3 times the acceptance vibration test level (g rms); the spectrum was the same as that of the acceptance tests. The qualification temperature levels to simulate acceptance test levels were defined to be 11 K (20° F) above and 11 K (20° F) below the acceptance test temperature range. (The acceptance qualification levels are shown in figures 5 and 6.) The vacuum level was defined as 1.333 mN/m^2 (1×10^{-5} torr) or less. The qualification test durations were as long as 5 times the acceptance test durations to allow for retests.

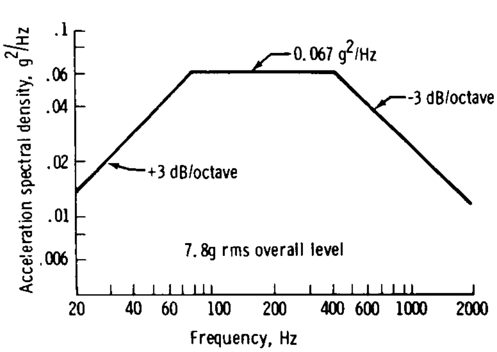


Figure 6. - Requalification requirements for Apollo minimum vibration acceptance testing.

Monitoring

Functional tests or continuity tests, or both, were conducted on all components before, during, and after the environmental acceptance tests. If complete functional verification was impossible during the acceptance tests, because of limited test time, then critical crew safety and mission success functions were given priority. All other circuits were continually monitored during the test for continuity and unwanted short circuits.

Retests

After all failures were repaired, the unit was subjected to a retest. The contractor was not authorized to grant waivers for acceptance tests. Also, the hardware was not to be accepted without the required acceptance retest unless a waiver had been granted by MSC. In no case was the accumulative acceptance test time, plus the anticipated mission time, permitted to exceed the qualification test time for that environment.

ENVIRONMENTAL ACCEPTANCE TESTING IMPLEMENTATION IN THE APOLLO PROGRAM

Several LM and Block II CSM spacecraft had completed assembly and were in checkout when the decision was made to implement the more rigorous environmental acceptance test program. Thus, only selected components were removed from these spacecraft for acceptance vibration testing. The effectivity for component selection was different on the early manned spacecraft because the spacecraft had already been assembled when the test program was initiated.

Vibration Test Criteria

The criteria used for component acceptance vibration test selection were as follows.

First manned CSM and LM. - For the first manned CSM and LM, only crew safety equipment was tested. A crew safety (Criticality I) component is one in which a failure by itself or in combination with an undetected failure could create an associated single failure point that could impair crew safety. Crew safety equipment was defined as that which, if disabled, could result in loss of abort capability, loss of caution and warning, loss of voice communication, inadvertent engine firing, loss of attitude control, or loss of an habitable environment. Provision of redundancy did not automatically remove equipment from the crew safety category because redundant equipment of like configuration could contain the same workmanship fault.

Second manned CSM and LM. - For the second manned CSM and LM, crew safety and mission success (Criticality I and II (primary objective)) equipment was tested. A mission success component is one in which a failure by itself could cause the loss of a mission or a primary objective.

Third manned CSM and LM and succeeding spacecraft. - For the third manned CSM and LM and succeeding spacecraft, all selected components (Criticality I, II, and III (secondary objective)) were tested. The list of components selected from all categories for acceptance vibration testing is contained in appendix C.

The acceptance vibration test criteria (fig. 1) in a number of cases exceeded the original qualification levels. Therefore, a significant quantity of LM and CSM hardware required requalification to the 7.8g rms spectrum shown in figure 6. Requalification was required on 19 of the 65 CSM components and 26 of the 83 LM components that were subject to acceptance vibration requirements. These components are identified in appendix C. In numerous cases, the acceptance test level was modified slightly to avoid the necessity of requalification and yet satisfy the intent of the new acceptance tests. An example of a component tested to modified levels is shown in figure 7. Totals of 39 of 83 LM components and 10 of 65 CSM components were tested to modified spectra.

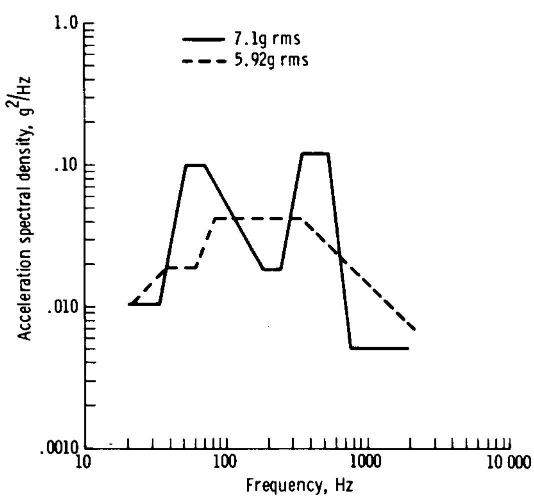


Figure 7. - Examples of modified vibration spectra.

Thermal/Thermal-Vacuum Test Criteria

The acceptance thermal/thermal-vacuum tests were implemented as an in-line function; however, all component replacements, including the earlier spacecraft, were to be made with units that had received acceptance thermal/thermal-vacuum tests. Flight usage of a component that had not received acceptance thermal/thermal-vacuum testing required that three like components had received acceptance thermal/thermal-vacuum testing before the mission. Using the acceptance test data from like components, the lot sampling technique was used in determining the flight acceptability of hardware that had not been tested.

The component selection criteria used for thermal/thermal-vacuum acceptance testing were based on the criticality of the hardware. The list of the selected components is contained in appendix C.

In some cases, the revised Apollo acceptance thermal/thermal-vacuum test requirements exceeded the qualification levels. To avoid the necessity of requalification, the temperature sweep (fig. 5) was reduced slightly from the optimum 56 K (100° F), and the differential temperature between acceptance and qualification extremes was reduced from 11 to 5.5 K (20° to 10° F) and, in one or two cases, to 2.8 K (5° F).

ENVIRONMENTAL ACCEPTANCE TEST RESULTS

A summary of the environmental acceptance test history is presented in tables II to IV and figures 8 to 11. These data were compiled from the test history of the environmental acceptance test program imposed after mid-1967.

Some 11 961 component tests were performed on 148 types of components during the acceptance vibration test program with a failure rate of 6.85 percent. Some 4286 component tests were performed on 126 types of components during the acceptance thermal/thermal-vacuum test program with a failure rate of 15.98 percent. The smaller number of thermal/thermal-vacuum tests was a result of the later effectivity of this test program. An overall accounting of the environmental acceptance testing performed on a selected number of component types is presented in table II.

TABLE II. - APOLLO SPACECRAFT ENVIRONMENTAL

ACCEPTANCE TEST HISTORY[a]

Acceptance test item	Number of components tested	Different component types	Failures	
			Total	Percent
Vibration				
CSM	5 613	65	221	3.94
LM	6 348	83	598	9.42
Total	11 961	148	819	6.85
Thermal vacuum				
CSM	1 179	55	158	13.40
LM	3 107	71	527	16.96
Total	4 286	126	685	15.98

[a]The data from which this table was developed were received from North American Rockwell Corporation and Grumman Corporation in monthly status reports.

TABLE III. - APOLLO SPACECRAFT ACCEPTANCE TEST HISTORY

[As of Sept. 1, 1970]

Subsystem	Vibration						Thermal/thermal vacuum					
	Number of units tested		Failures				Number of units tested		Failures			
			Workmanship		Design				Workmanship		Design	
	CSM	LM	CSM	LM	CSM	LM	CSM	LM	CSM	LM	CSM	LM
Propulsion	39	244	0	9	0	5	8	17	0	1	0	0
Reaction control	--	584	--	8	--	0	112	672	0	47	0	0
Sequencers	168	--	6	--	0	--	165	--	5	--	1	--
Mechanical explosive	--	108	--	9	--	4	--	80	--	9	--	0
Environmental control	184	189	2	8	8	5	131	97	6	4	2	1
Crew provisions	--	61	--	3	--	2	--	17	--	3	--	0
Displays and controls	3901	4181	7	227	47	32	8	1184	0	142	0	63
Instrumentation	33	137	0	19	0	2	7	461	0	23	0	0
Communications	336	139	26	26	14	10	204	96	14	11	15	3
Electrical power	466	309	23	14	13	6	156	307	3	10	5	7
Guidance and control[a]	484	396	14	55	8	19	388	177	23	27	21	40

[a]Includes radar subsystem on LM and stabilization and control subsystem on CSM.

TABLE IV. - SAMPLES OF DEFECTS DISCLOSED BY ENVIRONMENTAL ACCEPTANCE TESTING

(a) Command and service module

Component	Failure	Test phase
Electronic control assembly	Defective module	During vibration
Flight director attitude indicator	Contamination	During vibration
Radiofrequency (rf) coaxial switch	Teflon chip on rf contact	During vibration
Antenna assembly	Coaxial line connectors backed off (epoxy not properly cured)	During vibration
Reaction control system control box	Wire improperly inserted in terminal board	During vibration
Mission events sequence controller	Insulating material between relay contacts	During thermal
Service module jettison controller	Premature time delay actuation	During thermal
Power factor correction	Break or nick in fuse wire	During thermal
Rotation controller	Damaged terminal and broken wire	During thermal
Thrust vector position servomechanism	Damaged wire insulation	After thermal
Electronic control assembly	Broken resistor	During thermal
Rotation controller	Pitch gear binding	During thermal
Signal-conditioning equipment	Damaged transistor	During thermal

TABLE IV. - Continued

(b) Lunar module

Component	Failure	Test phase
Descent engine control assembly	Dewetted solder joint	During vibration
Attitude translation control assembly	Defective solder joint on diode	During vibration
Attitude translation control assembly	No solder at joint with cordwood	After vibration
Abort control assembly	Pitch drive shaft not inserted far enough into clamp	After vibration
Abort electronics assembly	Intermittently open capacitor	During vibration
Abort sensing assembly	Collector leads broken on transistor	After vibration
Rendezvous radar electronics assembly	Relay contamination	After vibration
Reaction control system solenoid valve	Potting not complete; glass fracture	After vibration
Reaction control system solenoid valve	Contamination on magnet faces	After vibration
Reaction control system solenoid valve	Contamination on Teflon seat	After vibration
Stabilization and control assembly	Relay contamination	After vibration
Caution and warning electronics assembly	Relay distortion prevented current flow	During vibration
Auxiliary relay switch assembly	Open relay coil	After vibration
S-band steerable antenna	Improper mating of male and female pins	During vibration

TABLE IV.- Concluded

(b) Concluded

Component	Failure	Test phase
S-band steerable antenna	Misalinement of windup mechanism	After vibration
Very-high-frequency transceiver	Intermittent relay contacts	After vibration
Rate gyro assembly	Faulty stator	During thermal vacuum
Abort control assembly	Improper calibration	During thermal vacuum
Abort control assembly	Improper centering of sector gear	During thermal vacuum
Reaction control system engine chamber pressure	Quality yield problem	During thermal
Lunar surface sensing probe	Reed switch failed	During thermal vacuum
Carbon dioxide sensor	Defective capacitor	During thermal
Stabilization and control assembly	Relay contamination	During thermal
Pressure transducer	Poor lead routing	After thermal
S-band power amplifier	Improper resistor selector	During thermal vacuum
Emergency detection relay box	Contamination	During thermal vacuum
Auxiliary switch relay box	Defective splice	During thermal
Inverter	Integrated circuit leakage	During thermal vacuum
Inverter	Broken wire (excess crimping)	During thermal vacuum
Floodlight	Broken wire in potting	During thermal

A comparison of the acceptance thermal/thermal-vacuum and vibration testing is presented in figure 8. Workmanship defects accounted for 7.65 percent of the thermal/thermal-vacuum test failures as compared with the 3.81 percent for the acceptance vibration tests. Although the purpose of environmental acceptance tests was to detect workmanship and manufacturing defects, a significant number of design errors were also detected. Design defects accounted for 3.68 percent of the thermal/thermal-vacuum test failures as compared with 1.46 percent of the vibration test failures. The number of workmanship and design failures disclosed by acceptance vibration and thermal/thermal-vacuum tests is presented by subsystem in table III. In table IV, samples of the defects disclosed by the environmental acceptance testing are presented with a notation showing the type of test that revealed the failure.

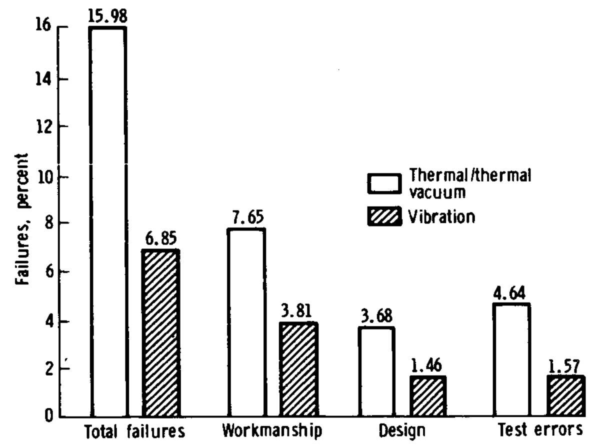
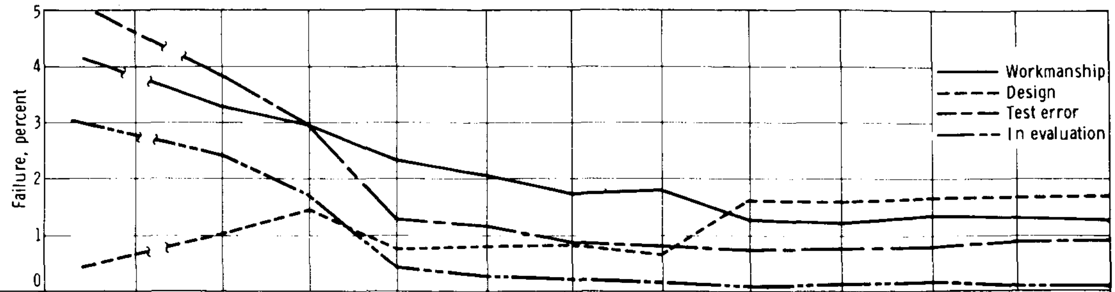


Figure 8. - Comparison of vibration and thermal failures during acceptance tests.

The failure trends throughout the environmental acceptance test program are presented in figures 9 to 11. The figures show the accumulative failure trends for workmanship flaws, design defects, test errors, and failures still in evaluation. In figure 9(a), during the period from July to September 1969, the marked increase in design failures was a result of the reevaluation and reclassification of a number of circuit breaker failures from workmanship to design. The increase in workmanship failures shown in figure 9(b) during the period from September 1968 to June 1969 was attributable, in part, to the increasing number of component types being subjected to acceptance vibration testing. The increase in thermal/thermal-vacuum failures shown in figures 10 and 11 resulted from additional types of components being integrated into the program. Finally, the failures caused by test errors remained at a level much higher than expected.

CONCLUSIONS AND RECOMMENDATIONS

Before mid-1967, very little emphasis was placed on environmental acceptance testing as a method of detecting defects in Apollo spacecraft hardware. Although rigorous environmental acceptance tests were implemented late, the tests were both comprehensive and effective. To provide an effective screen for workmanship and manufacturing defects, environmental acceptance tests must have minimum levels to which the hardware will be subjected. These minimum levels must be established independently of flight levels and conditions.

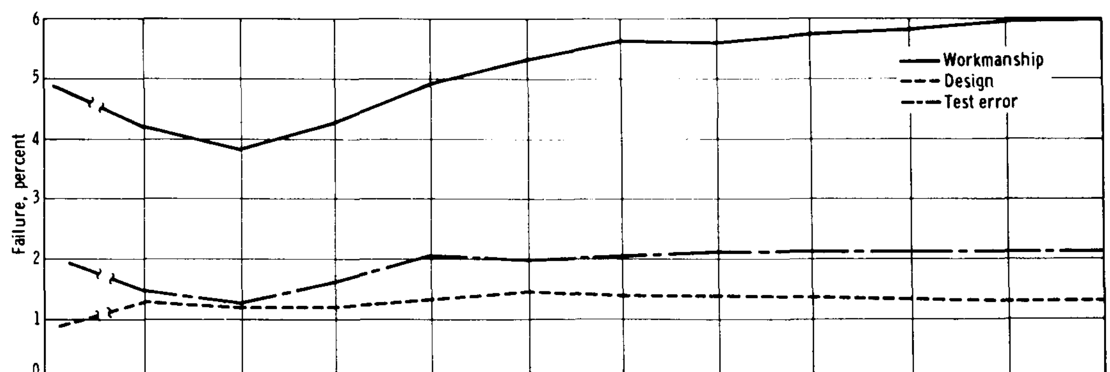


Parameters	1967	1968				1969				1970		
	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.
No. of units tested	246	493	641	1947	2602	3606	4584	5510	5587	5607	5611	5613
Workmanship failures	(a)	16	19	46	54	64	82	[b]66	66	76	76	76
Design failures	(a)	5	10	15	21	29	30	[b]85	86	88	88	89
Test errors	(a)	19	19	25	29	30	34	41	43	49	53	53
In evaluation	(a)	12	11	10	8	9	10	3	2	6	2	2
Total failures	37	52	59	96	112	132	156	195	197	219	219	220

[a]No breakdown of data during this time frame.

[b]Circuit breaker failures reevaluated and changed from workmanship to design.

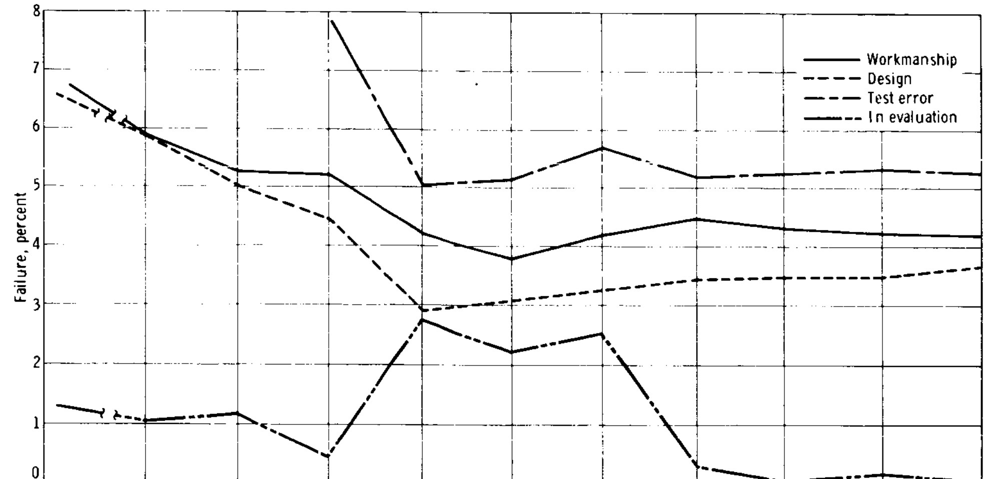
(a) CSM.



Parameters	1968				1969				1970		
	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.
No. of units tested	2399	3887	4462	4966	5377	5707	6074	6183	6331	6345	6348
Workmanship failures	100	149	190	240	285	324	341	353	367	376	378
Design failures	33	49	56	69	81	85	85	85	85	85	85
Test errors	34	50	75	101	107	118	129	131	133	134	135
Total failures	167	248	321	410	473	527	555	569	585	595	598

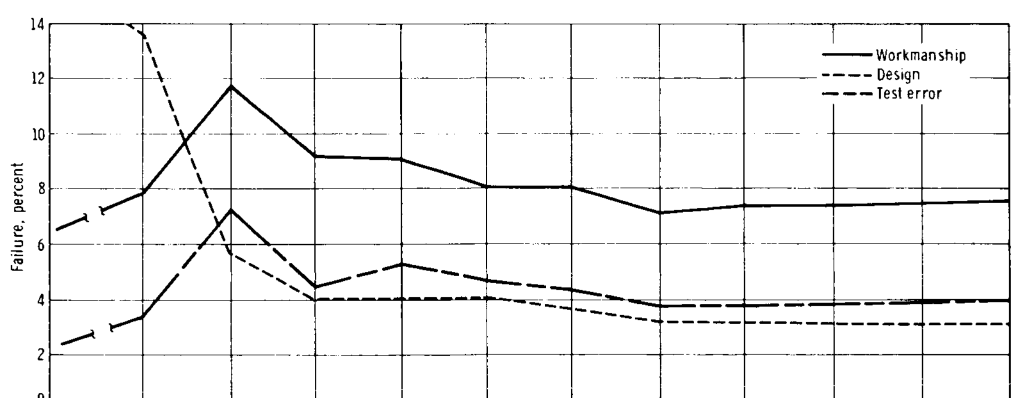
(b) LM.

Figure 9.- Acceptance vibration test failure trends.



Parameters	1968			1969				1970		
	Jan.-Oct.	Nov.	Dec.	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.
No. of units tested	288	337	417	710	850	925	1112	1166	1170	1179
Workmanship failures	17	18	22	30	33	39	50	51	51	51
Design failures	17	17	19	21	27	30	39	42	42	44
Test errors	25	28	33	36	44	53	58	62	63	63
In evaluation	3	4	2	20	19	24	3	0	2	0
Total failures	62	67	76	107	123	146	150	155	158	158

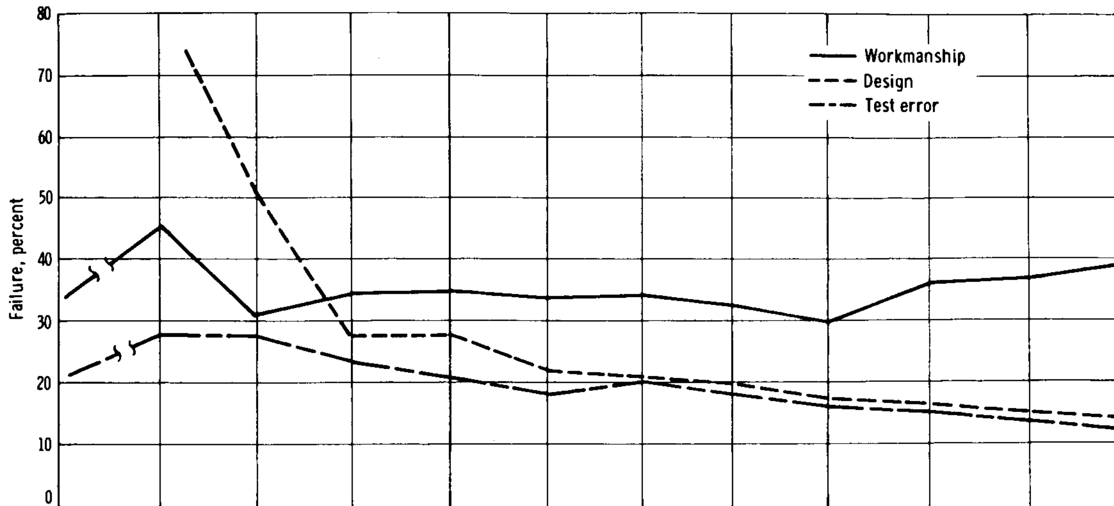
(a) CSM.



Parameters	1968				1969				1970		
	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.
No. of units tested	141	555	1270	1595	1999	2417	2828	2887	2971	2978	2982
Workmanship failures	11	65	117	144	165	194	206	215	222	225	227
Design failures	19	31	51	65	83	91	94	95	95	95	95
Test errors	5	40	57	86	97	107	107	110	113	116	118
Total failures	35	136	225	295	345	392	407	420	430	436	440

(b) LM.

Figure 10. - Acceptance thermal-vacuum test failure trends.



Parameters	1968				1969				1970		
	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.	Oct.-Dec.	Jan.-Mar.	Apr.-June	July-Sept.
No. of units tested	11	22	45	63	76	85	93	101	107	115	125
Workmanship failures	5	7	16	23	26	30	31	31	40	44	50
Design failures	9	11	13	18	18	18	19	19	19	19	19
Test errors	3	6	11	14	14	18	18	18	18	18	18
Total failures	17	24	40	55	58	66	68	68	77	81	87

Figure 11.- Acceptance thermal test failure trends for LM panel-level assemblies.

Based on the Apollo experience, the following recommendations are made for future space programs.

1. Formal environmental acceptance test requirements should be imposed early in the program. These requirements should be imposed early in the design stage to ensure that proper tests can be conducted and that adequate monitoring of hardware response during the test can be accomplished.

2. Environmental acceptance tests should be conducted at a specific level, equal to or greater than an established minimum level, that provides an effective screen for workmanship and manufacturing defects. This level should not be established as a percentage of the qualification level. Because the purpose of the environmental acceptance test is to screen for workmanship and manufacturing defects, it is logical that all components should be capable of withstanding the same environmental level. Therefore, the environmental acceptance levels should be considered when specifying qualification levels on future programs.

3. A study to determine optimum environmental test levels should be conducted. The Apollo Program used a specified minimum level or the flight environment level, whichever was greater, as the criterion for acceptance testing of hardware. A study should be conducted to determine whether a more effective level can be established for future programs.

4. For an effective test program, more rigorous test discipline should be enforced. As an example, of the 11 961 units acceptance vibration tested on the Apollo Program, 22.9 percent (188) of the 819 failures resulted from test errors. Of the 4286 units acceptance thermal/thermal-vacuum tested, 29.1 percent (199) of the 685 failures resulted from test errors.

Lyndon B. Johnson Space Center
National Aeronautics and Space Administration
Houston, Texas, April 1, 1976
914-89-00-00-72

APPENDIX A

INDUSTRIAL SURVEY OF ACCEPTANCE VIBRATION TESTING

INTRODUCTION

This appendix contains a summary of the data obtained from the industrial survey conducted as a result of the wide variation in the acceptance vibration test requirements among the NASA centers and programs. The results of the survey, made in October 1967, were used to establish confidence in the new acceptance vibration requirements for the Apollo Program. The spacecraft programs and vehicles considered and surveyed were as follows.

1. Ranger
2. Mariner
3. Biosatellite
4. Orbiting Geophysical Observatory (OGO)
5. Vela (nuclear detection satellite)
6. Pioneer
7. Surveyor
8. Early Bird
9. Applications Technology Satellite (ATS)
10. Syncom
11. Burner II
12. Lunar Orbiter
13. Environmental Science Service Administration (ESSA)
14. Relay
15. Space electric rocket test (SERT)
16. Tiros
17. Mercury
18. Gemini

19. Nimbus

20. Agena payloads

In most of the programs surveyed, the components were subjected to random vibration acceptance testing, with the exceptions of the Biosatellite, OGO, Vela, Pioneer, and ATS programs. In these programs, sinusoidal vibration acceptance testing was used, with peak levels of ±5g. Some acceptance vibration tests were conducted at the spacecraft level. The spacecraft programs surveyed, the test levels, and the qualification factors are presented in table A-I.

TABLE A-I. - SPACECRAFT PROGRAMS SURVEYED, TEST LEVELS, AND QUALIFICATION FACTORS

Program/vehicle	Spacecraft weight, kg (lb)	Random test level, g rms	Sine only	Qualification factor, Qualification g rms / Acceptance g rms
Ranger	363 (800)	7.9	--	1.78
Mariner	261 (575)	9.0	--	1.82
Biosatellite	431 (950)	--	X	1.56
OGO	522 (1150)	--	X	1.50
Vela (nuclear detection satellite)	220 (485)	--	X	1.39
Pioneer	66 (145)	--	X	1.55
Surveyor	1043 (2300)	4.5	--	1.50
Early Bird	41 (90)	[a]6.5	--	1.41
ATS	340 (750)	--	X	1.41
Syncom	36 (80)	[a]6.5	--	1.41
Burner II	113 (250)	5.9	--	3.16
Lunar Orbiter	386 (850)	17.2	--	1.19
ESSA	139 (307)	6.2	--	1.50
Relay	81 (178)	7.7	--	1.53
SERT	170 (375)	7.7	--	1.53
Tiros	129 (285)	7.0	--	3.00
Mercury	1225 (2700)	7.6	--	1.83
Gemini	3402 (7500)	6.2	--	1.42
Nimbus	590 (1300)	9.2	--	1.50
Agena payloads	--	12.0	--	1.41

[a]Spacecraft level testing used for small satellites.

COMPONENT TESTING

Qualification and acceptance testing was conducted at the component level and at the system level in most of the programs. In a number of programs, a selected number of components were tested at the component level, followed by spacecraft level testing. In the Early Bird and Syncom programs, vibration acceptance tests were conducted at the spacecraft level only. The qualification and acceptance testing at the component level was conducted with the test article mounted to the vibration source in a manner simulating its flight installation. In general, the acceptance vibration test levels and spectra used were based on the expected mission environments for the particular piece of hardware. The components were not operated during vibration acceptance testing except when the hardware was required to operate in this type of environment during flight. The acceptance vibration g rms levels and qualification factors given in table A-I indicate the wide variations among programs.

Vibration Level Comparison

A comparison of the Apollo minimum levels and spectra and those of the surveyed programs is shown in figure A-1. The spacecraft programs included in this comparison had a maximum vibration acceptance level of 12.0g rms and a minimum level of 4.5g rms. The average level of the programs surveyed was 8.8g rms as compared to the Apollo minimum level of 6.1g rms. Programs included in the survey were Ranger, Agena, Burner II, Mariner, Nimbus, Gemini, and Mercury. The Lunar Orbiter was omitted because the acceptance test level was too high for consideration.

Table A-II is a comparison of the Apollo minimum level with those of a number of the spacecraft programs surveyed in the 20- to 400-hertz range. The Apollo minimum of 3.75g rms is approximately midway between the high of 5.16g rms and the low of 1.82g rms. A comparison of the overall Apollo minimum g rms level and those of the surveyed programs is shown in figure A-2, with the Apollo minimum level being slightly below the average.

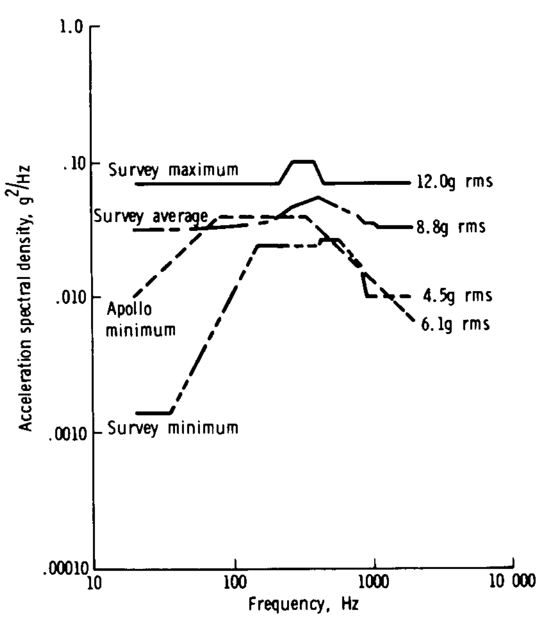


Figure A-1. - Random vibration acceptance test levels.

Failure Detection Experience

A detailed review of the failures experienced on the Surveyor program, on the Lunar Orbiter program, and on several NASA Goddard Space Flight Center (GSFC) managed unmanned spacecraft programs is summarized in figure A-3. In each of these programs, the hardware was both vibration and

TABLE A-II. - RANDOM VIBRATION ACCEPTANCE

TEST REQUIREMENTS

Program	Level, g rms	
	20 to 400 Hz	Total spectrum
Ranger	3.90	7.9
Agena	3.08	10.3
Burner II	2.83	5.9
Mariner	3.94	9.0
Nimbus	5.16	11.2
Gemini	3.42	6.6
Mercury	4.93	7.6
Lunar Orbiter	1.82	17.2
Apollo minimum	3.75	6.1

thermal-vacuum acceptance tested. For the GSFC spacecraft programs, only a certain number of components were acceptance tested at the component level. During the other two programs, all the components were acceptance tested at the component level before being subjected to the spacecraft level acceptance testing. It should be noted that the spacecraft level thermal-vacuum testing conducted on these three programs disclosed more defects than the spacecraft level vibration testing.

During the Lunar Orbiter environmental acceptance testing at the component level, 54 faults were disclosed in 256 vibration tests and 27 faults were disclosed in 250 thermal-vacuum tests. An analysis of these failures revealed that, of the 54 vibration failures, 33 were mechanical; 14, electronic; 6, electrical; and 1, structural. Of the 27 thermal-vacuum failures, 9 were mechanical; 13, electronic; and 5, electrical.

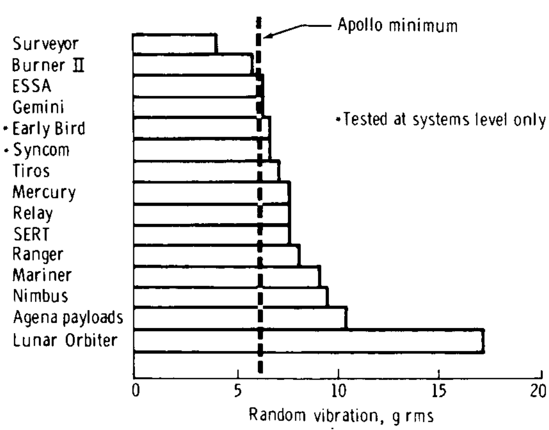


Figure A-2. - Acceptance test levels.

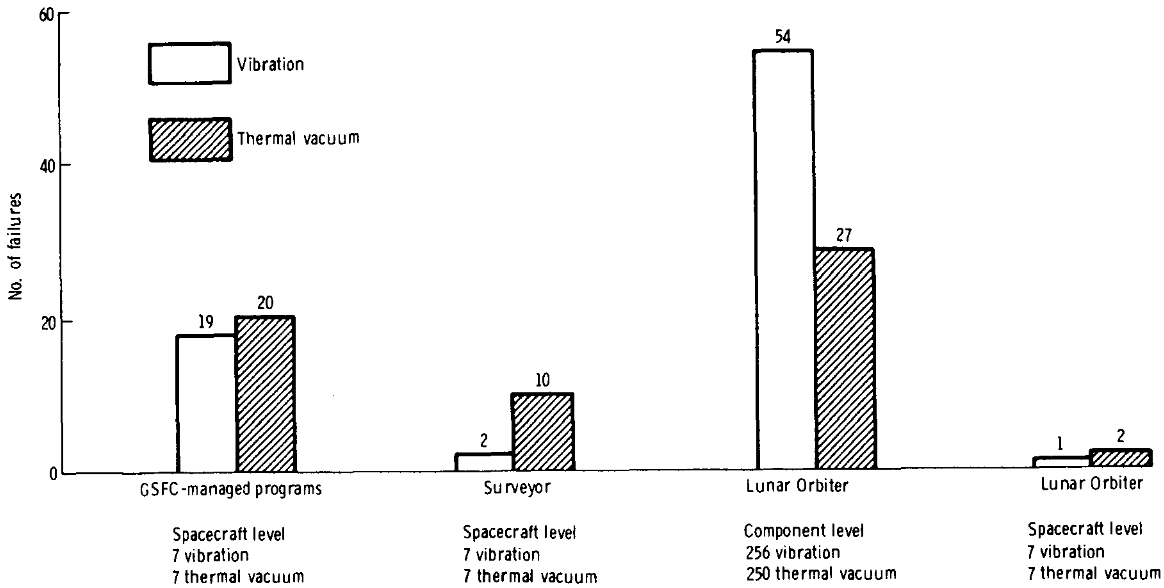


Figure A-3. - Failure detection experience.

The Lunar Orbiter environmental acceptance testing failures can be placed in the following four categories.

Category	Vibration acceptance	Thermal-vacuum acceptance
Workmanship	8	5
Manufacturing	5	5
Part failure	5	2
Design inadequacy	36	15

SURVEY RESULTS

The following specific conclusions were drawn from this survey.

1. The selected Apollo minimum level g rms was slightly below average with respect to the programs surveyed.

2. With the exception of two, all the programs reviewed used a higher acceptance vibration level than the Apollo Program minimums.

3. The acceptance vibration test levels for the programs surveyed were normally based on expected mission levels.

4. Most equipment was operated during acceptance vibration testing only when the item was expected to operate in a vibrating environment during flight.

5. The qualification factors ranged from a low of 1.19 to a high of 3.16, compared to the Apollo factor of 1.3.

6. Thermal/thermal-vacuum acceptance testing is also required to provide an adequate screen to ensure the quality of the hardware.

APPENDIX B

INDUSTRIAL SURVEY OF ACCEPTANCE THERMAL/THERMAL-VACUUM TESTING

INTRODUCTION

An industrial survey was conducted in December 1967 to obtain background and supporting data for evaluating the Apollo thermal/thermal-vacuum test practices and establishing new thermal/thermal-vacuum requirements for the Apollo spacecraft. The following space vehicles and programs were surveyed.

1. Surveyor
2. Syncom
3. Applications Technology Satellite (ATS)
4. Orbiting Geophysical Observatory (OGO)
5. Pioneer
6. Intelsat III
7. Nimbus
8. Biosatellite
9. Lunar Orbiter
10. NASA Goddard Space Flight Center (GSFC) Agena payload
11. Burner II
12. Orbiting vehicle (OV-1)
13. Mariner

Generally, components were subjected to both qualification and acceptance tests, with the exception of the Burner II and OV-1 programs. In these two programs, funding was limited and maximum use of previously qualified components was made. Consequently, qualification and acceptance tests were conducted only on components of new design. In the OV-1 program, only the first two flight vehicles were acceptance tested.

Detailed data for the GSFC payloads flown on the Atlas-Agena, Thor-Agena, and Delta-Agena launch vehicles were not obtained. However, most of these components were acceptance tested at anticipated mission temperature levels, and the qualification test levels were 8 K (15° F) higher and lower than the acceptance test range.

COMPONENT TESTING

Qualification and acceptance testing at the component level involved controlling the environment of the test article in a test chamber and recording its performance. Generally, for test articles containing internally mounted components, the test article was mounted on a test fixture and the temperature extremes were measured at the mounting surface. The test articles were operated in their simulated mission environment and the performance recorded.

The component acceptance and qualification test temperatures for various programs are summarized in figure B-1. The unshaded portion of the bars represents the acceptance test temperature limits, and the shaded portion of the bars represents the qualification temperature margins. Considerable variation existed in both the acceptance and qualification temperatures among programs. However, the average acceptance test temperature range for all the programs was from 273 to 314 K (32° to 105° F). The average qualification test temperature range was from 260 to 326 K (8° to 127° F), 12 K (22° F) above and 13 K (24° F) below the acceptance temperature levels. Figure B-2 shows the acceptance temperature range of the programs reviewed. The average temperature sweep was approximately 41 K (73° F), whereas the adopted Apollo acceptance test temperature sweep was 56 K (100° F).

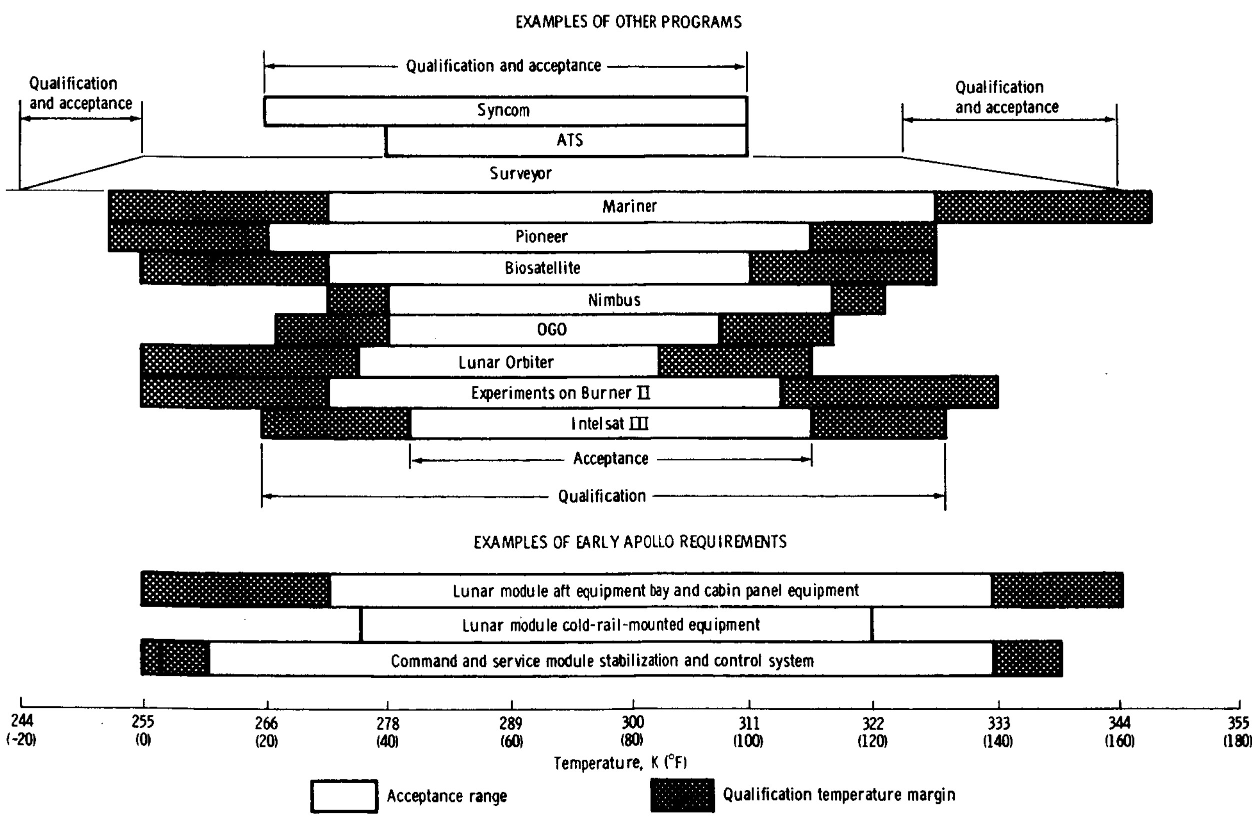


Figure B-1.- Thermal acceptance and qualification temperature limits.

The length of time that a component was maintained at the acceptance test temperature extreme varied from 30 minutes to 60 hours or to "sufficient time to reach steady state." Results from the Mariner program indicated that electronic equipment is much more susceptible to failure at high temperatures. Therefore, a steady-state condition was maintained 8 to 12 times longer at the upper temperature limit than at the lower temperature limit. Approximately 90 to 95 percent of the failures occurred during the first 12 days of qualification testing at the upper temperature limit. Therefore, for Mariner qualification testing, the component was maintained at 348 K (167° F) for 12 days.

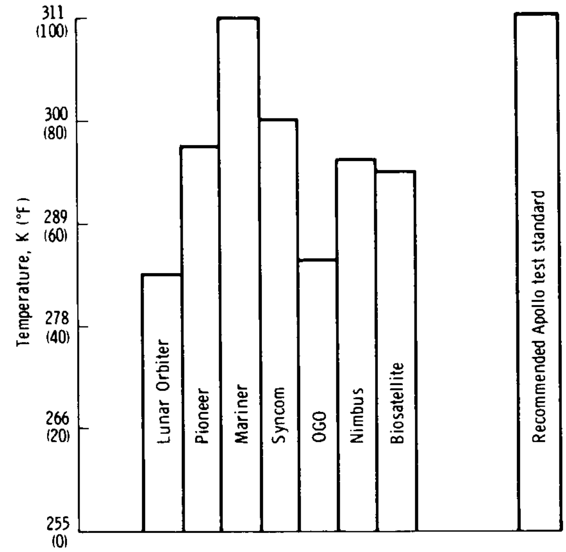


Figure B-2. - Industrial practice for thermal acceptance testing.

The vacuum chamber pressure was probably the most consistent value in the total thermal/thermal-vacuum test requirements. Nearly all areas surveyed specified a value of 1.333 mN/m^2 (1×10^{-5} torr) or less (table B-I), but two programs specified 0.1333 mN/m^2 (1×10^{-6} torr). In all cases, the test article was operating during the entire test, including chamber pumpdown.

SYSTEM TESTING

Complete integrated system tests generally consisted of placing the spacecraft in a vacuum chamber that had the capability of simulating the expected thermal-vacuum environment. The environment included a pressure of 1.333 mN/m^2 (1×10^{-5} torr) or less and a simulation of the external thermal environment. The two most common methods used for thermal simulation were to simulate the average environment sink temperature by means of zone panels along the chamber walls and to simulate the environment extremes by means of solar simulators and liquid-nitrogen-cooled chamber walls. During spacecraft testing, the normal modes of operation were verified and component temperatures were monitored.

For spacecraft qualification testing, self-induced heating and the worst-case combination of environmental extremes (maximum or minimum solar constant, maximum or minimum coating degradation, and maximum or minimum planet temperature and albedo) were used generally as the stimuli in the test. Component temperatures and system performance were monitored during these tests. The temperatures of flight components were not allowed to exceed the qualification temperature limits.

TABLE B-I. - INDUSTRIAL SURVEY VACUUM LEVELS

Program/vehicle	Vacuum, mN/m^2 (torr)	Test method	
		Solar simulation	Heaters
Surveyor	0.1333 (1×10^{-6})	X	--
Syncom	.1333 (1×10^{-6})	--	X
ATS	(a)	X	--
OGO	1.333 (1×10^{-5})	X	--
Pioneer	1.333 (1×10^{-5})	X	--
Intelsat III	1.333 (1×10^{-5})	X	--
Nimbus	1.333 (1×10^{-5})	(a)	(a)
Biosatellite	1.333 (1×10^{-5})	--	X
Lunar Orbiter	1.333 (1×10^{-5})	X	--
MSFC[b] Agena payload	(a)	X	--
OV-1	1.333 (1×10^{-5})	(a)	(a)
Mariner	1.333 (1×10^{-5})	X	--

[a]Unknown.

[b]NASA George C. Marshall Space Flight Center.

Nominal design environment and self-generated heat were used as the stimuli for acceptance testing. The test article performance and temperature were monitored while it was operated in all its modes.

The duration of the spacecraft level testing varied from program to program. However, the two dominant approaches for determining test duration were calculated time to reach steady state (used when simulating the average space sink temperature levels) and the time equivalent to three orbits (used when simulating the solar spectrum) to obtain the dynamic effects of entering and exiting from the shadow of the planet.

SURVEY RESULTS

The following specific conclusions were drawn from this survey.

1. A margin of approximately 13 K (23° F) between the acceptance test temperature levels and the qualification test temperature levels occurred.

2. The average acceptance test temperatures were from 273 to 314 K (32° to 105° F), with the exceptions of the Mariner and Lunar Orbiter.

3. Vacuum chamber pressure was 1.333 mN/m^2 (1×10^{-5} torr) or less.

4. The equipment was operating during the test. The time at steady-state levels and the number of temperature cycles to which components were exposed varied widely among the programs.

APPENDIX C
ACCEPTANCE TESTING COMPONENT LIST

TABLE C-I. - VIBRATION TESTS COMPONENT LIST

(a) Command and service module (CSM)

Component	Part no.	Increased qualification	CSM effectivity			
			101	103	104	106 and subsequent
Sequencers						
Master events sequence controller	ME901-0567-0019		X	X	X	X
Service module (SM) jettison controller	ME901-0569-0012		X	X	X	X
Lunar docking events controller	ME476-0035-0001			X	X	X
Lunar module (LM) separation sequence controller	ME450-0007-0001				X	X
Pyro continuity verification box	V16-540130-201		X	X	X	X
Environmental control subsystem (ECS)						
Water/glycol (W/G) flow-proportioning valve controller	ME476-0041-0001			X	X	X
Heater controller	ME476-0042-0002			X	X	X
W/G flow-proportioning valve	ME284-0331-0001			X	X	X
Cabin temperature control	ME284-0335-0001	X	X	X	X	X
Environmental control unit	ME901-0737		X	X	X	X
Cabin temperature controller	830010-4			X	X	X
Transducer		X	X	X	X	X
Power supply valve		X	X	X	X	X

TABLE C-I. - Continued

(a) Continued

Component	Part no.	Increased qualification	CSM effectivity			
			101	103	104	106 and subsequent
Stabilization and control subsystem (SCS)						
Flight director attitude indicator (FDAI)	ME432-0168-0202				X	X
Gyro assembly	ME493-0010-0102				X	X
Translation controller	ME901-0702-0002				X	X
Attitude-set control panel	ME901-0703-0102				X	X
Rotation controller	ME901-0704-0002				X	X
Electronic control assembly	ME901-0705-0202				X	X
Reaction jet and engine on-off controls	ME901-0706-0102				X	X
Gyro display coupler	ME901-0707-0002				X	X
Gimbal-position and fuel-pressure indicator	ME432-0167-0102				X	X
Thrust vector position servoamplifier	ME901-0708-0102				X	X
Electronic display assembly	ME901-0710-0202		X	X	X	X
Automated control						
Entry monitor system	ME432-0129				X	X
Instrumentation						
Instrumentation junction box	V36-759522				X	X
Power control module	V36-759525 and 3V36-759548				X	X

TABLE C-I. - Continued

(a) Continued

Component	Part no.	Increased qualification	CSM effectivity			
			101	103	104	106 and subsequent
Spacecraft junction box	V36-759560				X	X
Displacement	3V36-759031				X	X
Communications						
Very-high-frequency (VHF) transceiver	ME478-0065-0003				X	X
vhf/amplitude modulation (AM) transmitter-receiver	ME478-0067-0005	X			X	X
vhf recovery beacon	ME478-0069-0003				X	X
Audio center equipment	ME473-0086-0003			X	X	X
Premodulation processor	ME478-0068-0003			X	X	X
vhf triplexer	ME456-0040-0001			X	X	X
Central timing equipment	ME456-0041-0030 MC456-0041		X	X	X	X
Up-data link equipment	ME470-0101-0001 MC490-0101	X		X	X	X
Pulse code modulation (PCM) telemetry equipment	ME901-0719-0004	X		X	X	X
Signal conditioner	ME901-0713-0013 MC901-0713			X	X	X
S-band power amplifier	ME478-0066-0003	X		X	X	X
Unified S-band equipment	ME478-0070-0003	X		X	X	X
High-gain-antenna control unit	ME450-0010-0003 MC481-0008	X		X	X	X
2-kMC antenna switch	ME452-0052-0111 MC452-005			X	X	X
High-gain-antenna electronics assembly	ME476-0039-0003			X	X	X
High-gain antenna assembly	ME481-0008-0003			X	X	X

TABLE C-I. - Continued

(a) Continued

Component	Part no.	Increased qualification	CSM effectivity			
			101	103	104	106 and subsequent
Electrical power subsystem						
Power factor correction box	V36-452000	X			X	X
Direct-current power control panel	V36-452020	X	X	X	X	X
Main circuit breaker panel	V36-452050	X	X	X	X	X
Uprighting box	V36-452170	X	X	X	X	X
Battery circuit breaker panel	V36-452200	X	X	X	X	X
Alternating-current power control panel	V36-454000	X			X	X
Fuel-cell shutoff	V36-451240				X	X
Inverter input motor switch assembly	V36-454050	X			X	X
Fuel-cell remote control switch panel	V37-451200				X	X
Power distribution box	V37-451230		X	X	X	X
Inverter	ME495-0001-0006		X	X	X	X
Electrical wiring						
SCS junction box	V36-441209			X	X	X
Suit current limiter panel assembly	V36-443223				X	X
Circuit utilization panel assembly	V36-442213	X		X	X	X
Electrical control box assembly, reaction control system (RCS)	V36-447545		X	X	X	X

TABLE C-I. - Continued

(a) Concluded

Component	Part no.	Increased qualification	CSM effectivity			
			101	103	104	106 and subsequent
Electrical control box assembly, service propulsion system (SPS)	V37-440030		X	X	X	X
Electrical control box assembly, cryogenic system	V37-444010			X	X	X
Cryogenic control panel assembly	V37-445010			X	X	X
Displays and controls						
Caution and warning (C&W) equipment	430-0006	X	X	X	X	X

TABLE C-I. - Continued

(b) Lunar module

Component	Part no.	Increased qualification	LM effectivity				
			2	3	4	5	6 and subsequent
Propulsion subsystem							
Descent-engine "D" junction box	270-00600	X					X
Ascent-engine bipropellant valve assembly	270-00500	X					X
Descent-stage propellant quantity gaging system (PQGS) unit	270-00009					X	X
Descent-stage PQGS sensors	270-00009					X	X
Solenoid-latching valve, descent and ascent stages	270-713					X	X
Rough combustion cutoff assembly	270-723	X	X	X			
Propellant-level detector	270-801					X	X
Solenoid-operated valve, descent and ascent stages	270-00822				X	X	X
Stabilization and control subsystem							
Rate gyro assembly	300-110				X	X	X
Descent-engine control assembly	300-130				X	X	X
Attitude and translation control assembly	300-140			X	X	X	X
Attitude controller assembly	300-190			X	X	X	X
Abort electronics assembly	300-330				X	X	X
Abort sensor assembly	300-370				X	X	X

TABLE C-I. - Continued

(b) Continued

Component	Part no.	Increased qualification	LM effectivity				
			2	3	4	5	6 and subsequent
Data entry and display assembly	300-390				X	X	X
Thrust/translation controller assembly	300-28800				X	X	X
Rendezvous radar electronics assembly	370-100	X			X	X	X
Rendezvous radar antenna assembly	370-200				X	X	X
Landing radar electronics assembly	370-300				X	X	X
Landing radar antenna assembly	370-400				X	X	X
Reaction control subsystem							
Propellant solenoid valve	310-403				X	X	X
Mechanical design							
Lunar surface probe assembly	320-201	X				X	X
Environmental control subsystem							
Fan motor	330-118	X			X	X	X
Transducer	330-130	X		X	X	X	X
Fan motor	330-102	X			X	X	X
Coolant recirculation assembly (with 218 switch)	330-290	X			X	X	X
Cabin switch	330-323				X	X	X

TABLE C-I. - Continued

(b) Continued

Component	Part no.	Increased qualification	LM effectivity				
			2	3	4	5	6 and subsequent
Crew provisions							
Tracking light	340-00011	X		X	X	X	X
Utility light	340-413			X	X	X	X
Displays and controls							
Push-to-talk switch	350-90	X			X	X	X
Helium temperature and pressure indicator	350-201	X				X	X
Time-delay helium pressure equipment	350-202				X	X	X
Attitude indicator	350-301				X	X	X
Gimbal angle sequencing transformation assembly (GASTA)	350-302				X	X	X
Cross-pointer meter	350-305					X	X
Range/rate indicator	350-307	X				X	X
CA1, CA2, and CA3 stabilization control panels	350-308				X	X	X
Digital event timer	350-310				X	X	X
Apollo mission clock	350-312			X	X	X	X
RCS quantity indicator	350-401	X				X	X
Dual vertical meter	350-801					X	X
Toggle switches	350-8x					X	X
Rotary switches	350-803					X	X
Flag indicator	350-804					X	X
Component caution indicator	350-806					X	X
Pushbutton switches	350-808			X	X	X	X

TABLE C-I. - Continued

(b) Continued

Component	Part no.	Increased qualification	LM effectivity				
			2	3	4	5	6 and subsequent
C&W indicators	350-809	X				X	X
Synchro transmitter	350-60600					X	X
Instrumentation							
PCM and timing electronics assembly	360-2				X	X	X
Signal-conditioner electronic assembly	360-5	X		X	X	X	X
C&W electronics assembly	360-8	X		X	X	X	X
Data storage electronics assembly	360-12	X			X	X	X
Propulsion quantity measuring device	360-628					X	X
Communications							
Digital uplink assembly	380-00060				X	X	X
S-band transceiver	380-00130				X	X	X
Signal processor assembly	380-00170	X		X	X	X	X
vhf transceiver and diplexer	380-00250				X	X	X
S-band power amplifier	380-00290				X	X	X
S-band steerable antenna	380-00330	X			X	X	X
Electrical power subsystem							
General-purpose inverter	390-6	X		X	X	X	X
Lighting control subassembly	390-9				X	X	X
Lightweight relay junction box	390-23				X	X	X

TABLE C-I. - Continued

(b) Continued

Component	Part no.	Increased qualification	LM effectivity				
			2	3	4	5	6 and subsequent
Deadface relay	390-24				X	X	X
Ascent-stage electrical control assembly (ECA)	390-25	X		X	X	X	X
Descent-stage ECA	390-26				X	X	X
Power sensor fuse assembly	390-21055			X	X	X	X
Panel III module assembly	390-28125					X	X
Panel VIII module assembly	390-28115					X	X
Panel XII module assembly	390-51025					X	X
ECS relay box	390-28151			X	X	X	X
Ascent-engine arming assembly	390-28155			X			
Panel II module assembly	390-51026					X	X
Utility light switch assembly	390-52058	X	X	X	X	X	X
Rough combustion cutoff relay assembly	390-52195	X	X	X			
Fuse assembly no. 1	390-53057				X	X	X
Descent-engine prevalve diode assembly	390-53082			X	X	X	X
Panel I module assembly	390-53122					X	X
Explosive device relay box	390-53152			X	X	X	X
Auxiliary switch relay assembly	390-53154				X	X	X

TABLE C-I. - Concluded

(b) Concluded

Component	Part no.	Increased qualification	LM effectivity				
			2	3	4	5	6 and subsequent
Power failure relay assembly	390-53155	X		X	X	X	X
Attitude and translation control assembly output load resistor	390-53165	X		X	X	X	X
Ascent-stage batteries	390-21000			X	X	X	X
Descent-stage batteries	390-22000			X	X	X	X

TABLE C-II. - THERMAL/THERMAL-VACUUM TESTS COMPONENT LIST

(a) Command and service module

Component	Part no.	CSM effectivity								
		106	107	108	109	110	112	113	114	116 and subsequent
Sequencers										
Master events sequence controller	ME901-0567-0019						X	X	X	X
SM jettison controller	ME901-0569-0012						X	X	X	X
Lunar docking events controller	ME476-0035-0001						X	X	X	X
LM separation sequence controller	ME450-0007-0001						X	X	X	X
Pyro continuity verification box	V16-540130-201				X	X	X	X	X	X
Earth landing system controller	ME901-0001-0001			X	X	X	X	X	X	X
Environmental control subsystem										
Water/glycol (W/G) flow-proportioning valve controller	ME476-0041-0002								X	X
Backpressure valve	829170-3					X	X	X	X	X
Power supply temperature sensor	836066-4							X	X	X
Transducer	837076-4	X	X	X	X	X	X	X	X	X
Transducer	836130-1	X	X	X	X	X	X	X	X	X
Oxygen panel transducer (mass flow)	836136-1	X	X	X	X	X	X	X	X	X

TABLE C-II. - Continued

(a) Continued

Component	Part no.	CSM effectivity								
		106	107	108	109	110	112	113	114	116 and subsequent
Stabilization and control subsystem										
Gimbal-position and fuel-pressure indicator	ME432-0167-0102	X	X	X	X	X	X	X	X	X
FDAI	ME432-0168-0102	X	X	X	X	X	X	X	X	X
Gyro assembly	ME493-0010-0102	X	X	X	X	X	X	X	X	X
Translation controller	ME901-0702-0102	X	X	X	X	X	X	X	X	X
Attitude-set control panel	ME901-0703-0102	X	X	X	X	X	X	X	X	X
Rotation controller	ME901-0704-0202	X	X	X	X	X	X	X	X	X
Electronic control assembly	ME901-0705-0202	X	X	X	X	X	X	X	X	X
Reaction jet and engine on-off controls	ME901-0706-0102	X	X	X	X	X	X	X	X	X
Gyro display coupler	ME901-0707-0002	X	X	X	X	X	X	X	X	X
Thrust vector position servoamplifier	ME901-0708-0102	X	X	X	X	X	X	X	X	X
Electronic display assembly	ME901-0710-0202	X	X	X	X	X	X	X	X	X
Automated control										
Entry monitor (EM) control assembly	ME432-0188-0003	X	X	X	X	X	X	X	X	X
EM scroll assembly	ME901-0725-0006	X	X	X	X	X	X	X	X	X

TABLE C-II. - Continued

(a) Continued

Component	Part no.	CSM effectivity								
		106	107	108	109	110	112	113	114	116 and subsequent
Instrumentation										
Power control module box	V36-759525 and V36-759548					X	X	X	X	X
Communications										
Digital up-data link equipment	ME470-0101				X	X	X	X	X	X
Central timing equipment	ME456-0041				X	X	X	X	X	X
High-gain-antenna electronics assembly	ME476-0039	X	X	X	X	X	X	X	X	X
S-band power amplifier	ME478-0066					X	X	X	X	X
vhf/AM transceiver	ME478-0067					X	X	X	X	X
Premodulation processor	ME478-0068					X	X	X	X	X
Signal-conditioning equipment	ME901-0713	X	X	X	X	X	X	X	X	X
PCM telemetry equipment	ME901-0719					X	X	X	X	X
Audio center equipment	ME473-0086					X	X	X	X	X
Unified S-band equipment	ME478-0070					X	X	X	X	X
High-gain-antenna assembly	ME481-0008								X	X
Digital ranging assembly	ME478-0082					X	X	X	X	X
Electrical power subsystem										
Static inverter	ME495-0001-0008			X	X	X	X	X	X	X

TABLE C-II. - Continued

(a) Continued

Component	Part no.	CSM effectivity								
		106	107	108	109	110	112	113	114	116 and subsequent
Power factor correction box	V37-440080				X	X	X	X	X	X
Alternating-current power control box assembly	V36-454000-301					X	X	X	X	X
Direct-current power control panel	V36-452020-301						X	X	X	X
Inverter input motor switch assembly	V36-454050-301						X	X	X	X
SM power distribution box assembly	V37-451230-101						X	X	X	X
Electrical wiring										
SCS junction box assembly	V36-442260 and V36-442300					X	X	X	X	X
Suit current limiter panel assembly	V36-442320						X	X	X	X
Cryogenic panel assembly	V37-445025						X	X	X	X
Electrical control box assembly, RCS	V36-447580					X	X	X	X	X
Electrical control box assembly, SPS	V36-440050					X	X	X	X	X
Electrical control box assembly, cryogenic system	V37-444020						X	X	X	X
Circuit utilization panel assembly	V36-442213					X	X	X	X	X
Cryogenic fan control system box assembly	V37-447580						X	X	X	X

TABLE C-II. - Continued

(a) Concluded

Component	Part no.	CSM effectivity								
		106	107	108	109	110	112	113	114	116 and subsequent
Service propulsion subsystem										
Gage system control	ME450-0008-0011					X	X	X	X	X
Reaction control subsystem										
CSM solenoid valve	ME284-0276									X
Displays and controls										
Caution detection unit	ME430-0006					X	X	X	X	X

TABLE C-II. - Continued

(b) Lunar module

Component	Part no.	LM effectivity						
		3	4	5	6	7	8	9 and subsequent
Propulsion subsystem								
PQGS control unit	270-00009				X	X	X	X
Rough combustion cutoff assembly	270-723	X						
Stabilization and control subsystem								
Rate gyro assembly	300-110					X	X	X
Gimbal drive actuator	300-170			X	X	X	X	X
Thrust/translation controller assembly	300-28800				X	X	X	X
Data entry and display assembly	300-390	X	X	X	X	X	X	X
Attitude controller assembly	300-190	X	X	X	X	X	X	X
Reaction control subsystem								
Pointing control system pitch control switch	310-651				X	X	X	X
Propellant solenoid valve	310-403							X
Mechanical design								
Lunar surface probe assembly	320-201			X	X	X	X	X

TABLE C-II. - Continued

(b) Continued

Component	Part no.	LM effectivity						
		3	4	5	6	7	8	9 and subsequent
Environmental control subsystem								
Pressure transducer	I-130 (part of 330-190)	X	X					
Carbon dioxide sensor	330-150	X	X	X	X	X	X	X
Suit circuit assembly	330-190						X	X
Coolant recirculation equipment	330-290						X	X
Cabin pressure switch	330-323	X	X	X	X	X	X	X
Suit loop switch	330-326				X			
Oxygen control module	330-390						X	X
Crew provisions								
Tracking light	340-00011		X	X	X	X	X	X
Displays and controls								
Helium temperature/pressure indicator	350-201							X
Attitude indicator	350-301		X	X	X	X	X	X
GASTA	350-302					X	X	X
Cross-pointer meter	350-305		X	X	X	X	X	X
Range/rate indicator	350-307	X	X	X	X	X	X	X

TABLE C-II. - Continued

(b) Continued

Component	Part no.	LM effectivity						
		3	4	5	6	7	8	9 and subsequent
CA1, CA2, and CA3 stabilization control panels	350-308		X	X	X	X	X	X
Digital event timer	350-310	X	X	X	X	X	X	X
Mission clock	350-312	X	X	X	X	X	X	X
RCS quantity indicator	350-401		X	X	X	X	X	X
Synchro transmitter	350-60600		X	X	X	X	X	X
D'Arsonval meter	350-801		X	X	X	X	X	X
Toggle switches	350-8X		X	X	X	X	X	X
Rotary switches	350-803		X	X	X	X	X	X
Flag indicator	350-804		X	X	X	X	X	X
Component caution indicator	350-806		X	X	X	X	X	X
Pushbutton switches	350-808		X	X	X	X	X	X
C&W indicators	350-809		X	X	X	X	X	X
Instrumentation								
Pressure transducer	360-601						X	X
Pressure transducer	360-606						X	X
Docking light switch	360-616		X	X	X	X	X	X
Pressure transducer	360-624						X	X

TABLE C-II. - Continued

(b) Continued

Component	Part no.	LM effectivity						
		3	4	5	6	7	8	9 and subsequent
Communications								
Digital uplink assembly	380-00060				X	X	X	X
S-band transceiver	380-00130	X	X	X	X	X	X	X
Signal processor assembly	380-00170		X	X	X	X	X	X
vhf transceiver and diplexer	380-00250	X	X	X	X	X	X	X
S-band power amplifier	380-00290	X	X	X	X	X	X	X
Electrical power subsystem								
General-purpose inverter	390-6	X	X	X	X	X	X	X
Lighting control subassembly	390-9	X	X	X	X	X	X	X
Relay junction box	390-23	X	X	X	X	X	X	X
Deadface relay	390-24				X	X	X	X
Ascent-stage ECA	390-25			X	X	X	X	X
Descent-stage ECA	390-26	X		X	X	X	X	X
Power sensor fuse assembly	390-21055 360-10025	X	X	X	X	X	X	X
ECS relay box	390-28151	X	X	X	X	X	X	X
Panel I module assembly	390-53122		X	X	X	X	X	X
Panel III module assembly	390-28125		X	X	X	X	X	X

TABLE C-II. - Concluded

(b) Concluded

Component	Part no.	LM effectivity						
		3	4	5	6	7	8	9 and subsequent
Panel VIII module assembly	390-28115		X	X	X	X	X	X
Panel XII module assembly	390-52160		X	X	X	X	X	X
Fuse assembly no. 1	390-53057 360-10027	X	X	X	X	X	X	X
Fuse assembly no. 2	390-53050 360-10026	X	X	X	X	X	X	X
Descent-engine prevalve diode assembly	390-53082	X	X	X	X	X	X	X
Explosive device relay box	390-53152	X	X	X	X	X	X	X
Auxiliary switch relay assembly	390-53154		X	X	X	X	X	X
Panel assembly I	390-58101		X	X	X	X	X	X
Panel assembly II	390-58102		X	X	X	X	X	X
Panel assembly III	390-58103		X	X	X	X	X	X
Panel assembly IVA	390-58104		X	X	X	X	X	X
Panel assembly IVB	390-58107		X	X	X	X	X	X
Panel assembly V	390-58105		X	X	X	X	X	X
Panel assembly VIII	390-58108		X	X	X	X	X	X
Panel assembly XI	390-58110		X	X	X	X	X	X
Panel assembly XII	390-58112		X	X	X	X	X	X
Panel assembly XIV/XV	390-58114		X	X	X	X	X	X
Panel assembly VI	390-58106		X	X	X	X	X	X

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